Research Article

Wing Design for HSUAV for Endurance Performance with Pitching Moment Constraint

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Abstract

Wing design for High Speed Unmanned Air Vehicle (HSUAV) for endurance performance is worked out. High loitering lift coefficient demands maximum lift with minimum drag. The design to meet this requirement is seen to result in large negative pitching moment that has to be balanced though deflected up elevons/ elevators, which is drag penalizing. In our approach, the maximum lift to drag ratio at increased value of lift coefficient with constraint on resulting pitching moment is worked out. The pitching moment is constrained to specified values and lift is incremented till the endurance criterion is met.

Keywords: Endurance performance, Elevons/ elevators, HSUAV

Nomenclature

A = Panel Area

$a_{i,j}$ = Influence coefficients (Influence of j_{th} panel on i_{th} control point)

b = Span

c = Local chord length

C_{D0} = Profile drag coefficient

C_{Di} = Induced drag coefficient

C_L =Lift coefficient

Cp =Pressure coefficient

ΔCp = Pressure difference coefficient

D = Induced drag

L = Lift

LIF = Lift increment factor

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M_{∞} = Mach number of free stream

M_x =Wing root bending moment

M_y = Pitching moment

N = Total number of panels on half of wing

x, y, z = Cartesian coordinates in geometric plane; x is chordwise, y is span wise and z is vertical

XC = x-distance of panel control point

U = Free stream velocity

a = Angle of attack (Alpha)

γ = Circulation

Suffix

i = i_{th} Panel control point

j = j_{th} Panel

AO = After optimization

Introduction

In our approach UAV design is aimed for high altitude long endurance compounded with high speed flying capability. For a specified wing loading, the wing geometry can be worked out for high speed application by studying the effect of parametric variations on aerodynamic coefficients.[1,2] After a substantial survey, a moderately swept wing of aspect ratio of about seven is taken for our studies. High speed flying at a moderately high altitude at Mach number of 0.7, and 0.35 Mach number for endurance performance is considered. While propeller propulsive system is best choice for low –medium altitudes, a low by-pass ratio turbo-fan is preferable for higher altitudes. Considering a jet propulsion for our application, the maximum endurance condition to be applied is $C_{Di} = C_{D0}$. Lift coefficient can be driven from aerodynamic loading to support the weight of plane which is governed by M_{∞}^2 C_L. In our case since the Mach number is halved, therefore the lift coefficient has to go up by four times then its value at Mach number of 0.7. From known value of C_{D0} and C_L, the design is started from a lower value of C_L than the value provided, since it is incremented and subjected to support camber that results in minimum induced drag. The value of C_{D0} is taken as 0.025. Considering a wing loading of 6-7KN/m^2, and flying at Mach number of 0.7 at around 8-9 km altitude would result in a requirement of lift coefficient of around 0.2, and for a moderately swept and tapered cambered wing the alpha is around 2-2.5 degrees. If the endurance speed is taken as 0.35 Mach, it would result in lift coefficient requirement of 0.6-0.7, and that the resulting alpha is around 7-8^0. With all these combinations, now we attempt our design process.

Constraint optimization technique is applied to the definition of camber lines for minimum induced drag for the specified value of lift and pitching moment constraint. Program is developed in Fortran language. Gfortran complier is used that is available in Fedora operating system of VMware. Following command complies and creates a binary ready for executions.

gfortran – o file name file name.f

Following command executes the program file:

./ file name<input file name

Resulting optimal warp is broken into twist and optimal camber. Value resulting from twist at root minus the twist at tip is used to off-load the alpha. Residual twist is merged with residual camber to define a new camber line. Further NACA 0006 aerofoil is superimposed on this camber and pressure distribution determined. A flow chart below summarizes the process.

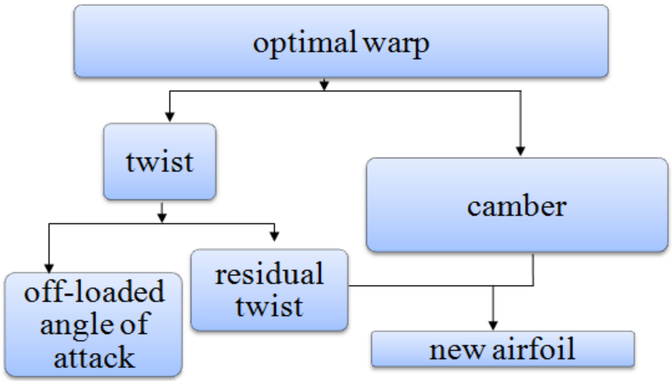

Figure 1.Flow chart of the process

Mathematical Modeling

In the approach made herein, wing is represented by a large number of constant pressure panels to model circulation.[3] These panels are used for estimation of pressure difference coefficient from where lift, induced drag and pitching moment are determined. Only half of the wing is considered. Other half of wing is imaged. The program uses vortex panel method to calculate downwash by satisfying tangential flow condition to determine singularity strength of each of panels. Control point for determining downwash is taken at 0.95% of local panel chord. The imaging of half of wing is done though the following expressions:

$$y(image) = -y_k - y_p \tag{1}$$

The subscript p refers to panel and subscript k refers to control point location. Panel circulation (γ) is determined through tangential flow boundary condition and pressure difference coefficient is given by $\Delta Cp = 2\gamma / U$. Lift is determined through integration of ΔCp over the chord and span. The program is developed in Fortran. It generates the output matrix of pressure difference coefficients from where lift, drag and moments are determined and then a optimization constraint of lift is introduced. Due to the effort of optimization, the given wing camber gets modified and hence a new pressure difference distribution results. Axis system and related moments are shown in Fig. 2. Half of wing is divided into nine chordwise and nine span wise panels, thereby making a total of 81 numbers of panels.

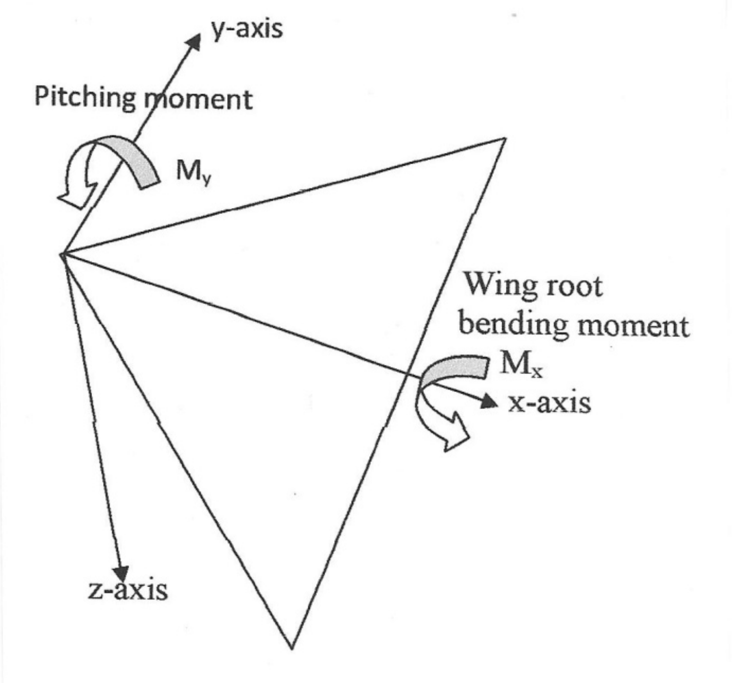

Figure 2.Axis system and related moments

Objective function (F) for the drag (D) minima and specified value of lift is written in the Lagrange form using Lagrange multipliers (λ_0)

$$F = D + \lambda_0 (L - \overline{L}) \tag{2}$$

Where $\overline{L}$ is the specified value of lift which is expressed in the following way:

$$\overline{L} = L \times LIF \tag{3}$$

Here L is the lift of a given profile (a flat wing is first considered i.e., the slopes of mean camber line are taken as zeros), which is to be increased by a factor LIF (LIF=1.0 corresponds to retaining lift to a value that corresponds to a flat plate case. Lift is expressed in terms of circulation as below for N number of panel singularities.[3]

$$L = \rho U\, [A_1 \gamma_1 + \ldots + A_N \gamma_N] \tag{4}$$

$$D = \rho U \begin{bmatrix} (a_{1,1}\gamma_1 + \ldots + a_{1,N}\gamma_N)\gamma_1 A_1 + (a_{2,1}\gamma_1 + \ldots + a_{2,N}\gamma_N)\gamma_2 A_2 \\ + \ldots\ldots\ldots\ldots + (a_{N,1}\gamma_1 + \ldots + a_{N,N}\gamma_N)\gamma_N A_N \end{bmatrix}$$

Here $\gamma_1, \ldots\ldots \gamma_N$ are circulation strength of N number of panels and $A_1, .. A_N$ are related panel areas. Lift effects are simulated by a spread of vortex sheet (circulation strength). The property of the vortex sheet is that the component of flow velocity tangential to sheet experiences a discontinuity change across the sheet that is given by:

$$\gamma = u_1 - u_2$$

Where u_1 and u_2 are tangential perturbation velocities just above and below the sheet respectively.

Matrix of optimization Eq. (6) below is obtained by

$$\begin{bmatrix} 2A_1 a_{1,1} & . & . & . & . & (A_1 a_{1,N} + A_N a_{N,1}) & A_1 \\ . & & & & & . & . \\ . & & & & & . & . \\ . & & & & & . & . \\ (A_N a_{N,1} + A_1 a_{1,N}) & . & . & . & . & 2A_N a_{N,N} & A_N \\ A_1 & . & . & . & . & A_N & 0 \end{bmatrix} \times \begin{bmatrix} \gamma_1 \\ . \\ . \\ . \\ \gamma_N \\ \lambda_0 \end{bmatrix} = \begin{bmatrix} 0 \\ . \\ . \\ . \\ 0 \\ \overline{L} \end{bmatrix}$$

differentiation of objective function (F) w.r.t circulation γ

Once the circulation is found, the warp to support the lift is determined from the following equation:

$$X_i = (a_{i,1}\gamma_1 + \cdots + a_{i,n}\gamma_n) A_i \tag{7}$$

In case the pitching moment is also to be used as a constraint, the objective function takes the following form[4,5]:

$$F = D + \lambda_0 (L - \overline{L}) + \lambda_1 (M_y - \overline{M}_y) \tag{8}$$

Where $\overline{L}$ the specified value of lift as explained before and $\overline{M}_y$ is constrained value of pitching moment i.e the value after optimization. M_y is the pitching moment of a given profile (a flat wing case). Matrix of optimization Eq.(9) below is obtained by differentiation of this objective function (F) w.r.t circulation γ.

$$\begin{bmatrix} 2A_1 a_{1,1} & . & . & . & . & (A_1 a_{1,N} + A_N a_{N,1}) & A_1 & A_1 X_1 \\ . & & & & & . & . & . \\ . & & & & & . & . & . \\ . & & & & & . & . & . \\ (A_N a_{N,1} + A_1 a_{1,N}) & . & . & . & . & 2A_N a_{N,N} & A_N & A_N X_N \\ A_1 & . & . & . & . & A_N & 0 & 0 \\ A_1 X_1 & . & . & . & . & A_N X_N & 0 & 0 \end{bmatrix} \times \begin{bmatrix} \gamma_1 \\ . \\ . \\ . \\ \gamma_N \\ \lambda_0 \\ \lambda_1 \end{bmatrix} = \begin{bmatrix} 0 \\ . \\ . \\ . \\ 0 \\ \overline{L} \\ \overline{M}_y \end{bmatrix}$$

Where $XC_I, \ldots XC_N$ are distances of panel control points from y-axis; and $a_{i,j}$ are influence coefficient towards downwash velocities.

Solution of this matrix for prescribed lift $\overline{L}$ and prescribed pitching moment $\overline{M}_y$ results in optimal circulation from where optimal warp, pressure difference coefficient, lift, drag, and pitching moment are determined.

Results and Discussions

The following wing is taken as candidate:

Leading edge sweep=20^0

Trailing edge sweep=9.2^0

Root chord=2.9

Tip chord=1.33

Span=16

Aspect ratio=7.5

Data is generated for a flat plate wing for alpha=8^0 and Mach number of 0.3. Wing is subjected to optimization with, and without the pitching moment constraint. Constraint on lift is maintained in all the cases.Table-1 summaries the data generated.

Table 1.Aerodynamic coefficients with and without pitching moment constraint

No.	Constraint	C_L	C_{Di} Before Optimization	C_{Di} After Optimization	% reduction in induced drag	$M_y / \overline{M}_y$	$M_x / M_{x,0}$
1	Lift	0.63	.0878	.0286	67%	.814	1.02
2	Lift and Pitching moment	0.63	.0878	.041	53%	1.0	1.2

Figure 3 shows the resulting chordwise camber distributions at y/b/2=0.5 station for these constraints. The maximum camber is 5.5% of local chord and is occurring at 45% of local chord in case of lift alone constraint. The % drag reduction is higher, however the pitching moment after optimization has increased, which is of concern. To overcome this increase, the value of pitching moment after optimization is maintained same as before optimization i.e., $M_y / \overline{M}_y = 1.0.$ The % drag reduction in this case has reduced. The resulting camber is shown in this Fig. 3. There is reflex in camber which is visible in this Figure which is plotted for y/b/2=0.5 station. In this case maximum camber is 5% at 30% of local chords. Ratio of wing root bending moment before and after optimization is seen to increase as a result of pitching moment constraint, which is in favor of decreased wing root bending moment.

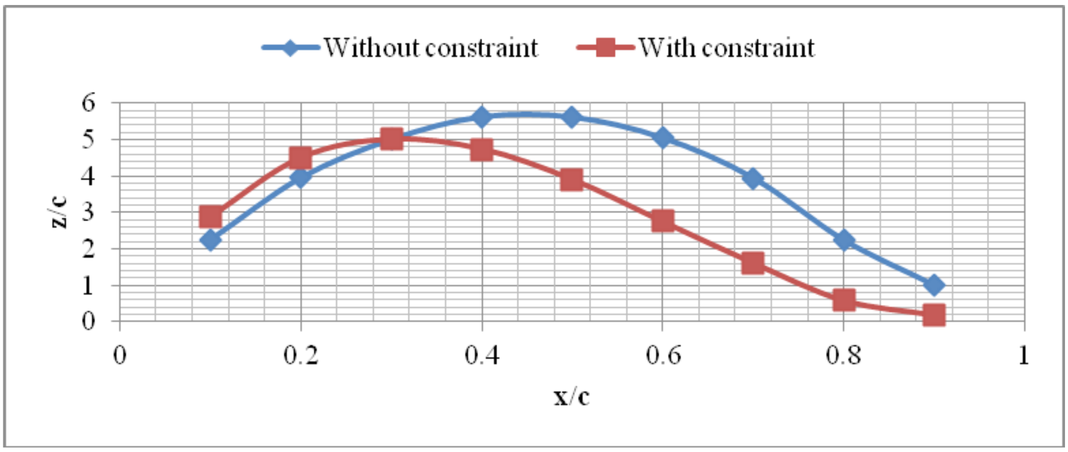

Figure 3.Chordwise camber distributions with and without pitching moment constraint at y/b/2=0.5 station

Now, since lift coefficient is to be incremented to reach design point, a lower value of alpha of 6^0 is taken with Mach number combination of 0.3. A code named `OPSGER` of reference- 4 is modified to do this. Optimisation is done starting from a flat plate wing, and data generated is shown in Table-2. In this case only lift constraint is used. Our design point is $C_L \approx 0.6$ and $C_{Di} \approx .025$ (which is the value of C_{D0}). Design point is reached at LIF=1.3, and is indicated by an arrow in Table-2. The resulting camber for the various values of LIF is shown in Figure 4. LIF increment has no

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effect on location of maximum camber which is occurring at 45% of chord. In all the cases max camber is occurring at 45% of local chords.

In case the design condition is not reached at around LIF value of 1.3, then the angle-of-attack should be increased to avoid excessive cambering of profile. Also if design condition is reached too early, then angle-of-attack should be reduced so as to generate reasonable camber. These aspects have the effect on the high speed lift coefficient value for the cruise alpha that is decided in the beginning, and the whole exercise can influence the geometric shape of wing.

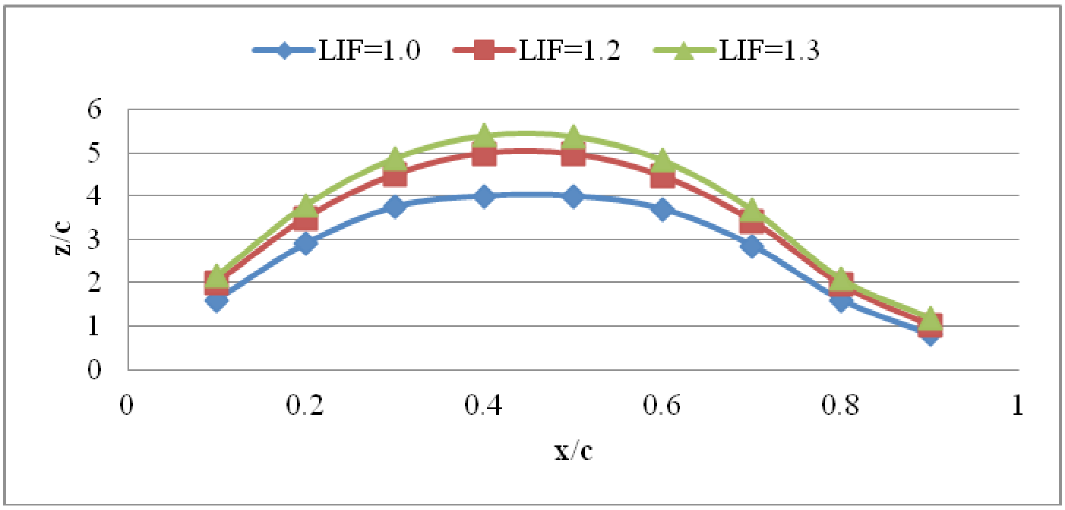

Figure 4.Camber distributions for various lift increment factors at y/b/2 station

Table 2.Aerodynamic coefficients without pitching moment constraint, lift is incremented though LIF

No.	LIF	C_L	C_{Di} after optimization
1	1.0	0.472	0.016
2	1.2	0.566	0.023
3	1.3	0.613	0.027
4	1.4	0.660	0.031

Now the pitching moment constraint is also imposed along with lift constraint, and data generated is shown in Table-3. The design point is seen not reachable, since at C_L≈0.6, C_{Di} ≈.099, which is much larger than the design specified value. Therefore, it is not advisable to use reflex cambered profiles for long endurance designs, and instead a canard could be used to overcome increased pitching moments.

Table 3.Aerodynamic coefficients with pitching moment constraint, lift is incremented though LIF

No.	LIF	C_L	C_{Di} after optimization
1	1.0	.4719	.023
2	1.2	.5663	.053
3	1.3	.6134	.099

Now Aerofoil NACA 0006 thickness is superimposed on the optimal camber lines of the design. The thickness solution for pressure distribution is obtained from a source-sink based panel code. Figure 5 shows the pressure distribution on the upper and lower surfaces at two stations, namely root and tip. Lower surface pressure distribution is unique, which appears flat all along the chord. This is because of the camber and the spanwise nature of flow which is peculiar to the sweep back and taper ratio combination of the wing. A large lift potential is visible by way of large distance between pressure on upper and lower surfaces. Similar plot for LIF=1.0 is shown in Fig. 6. There is similarity between these two. In the later case the area under the plot is smaller.

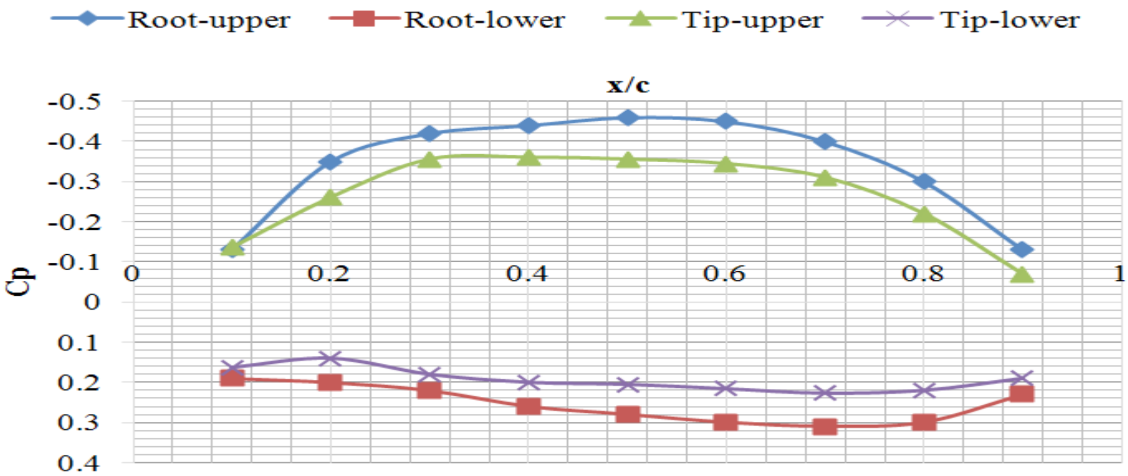

Figure 5.Pressure coefficients on upper and lower surfaces for the case of LIF=1.3 (Design Case)

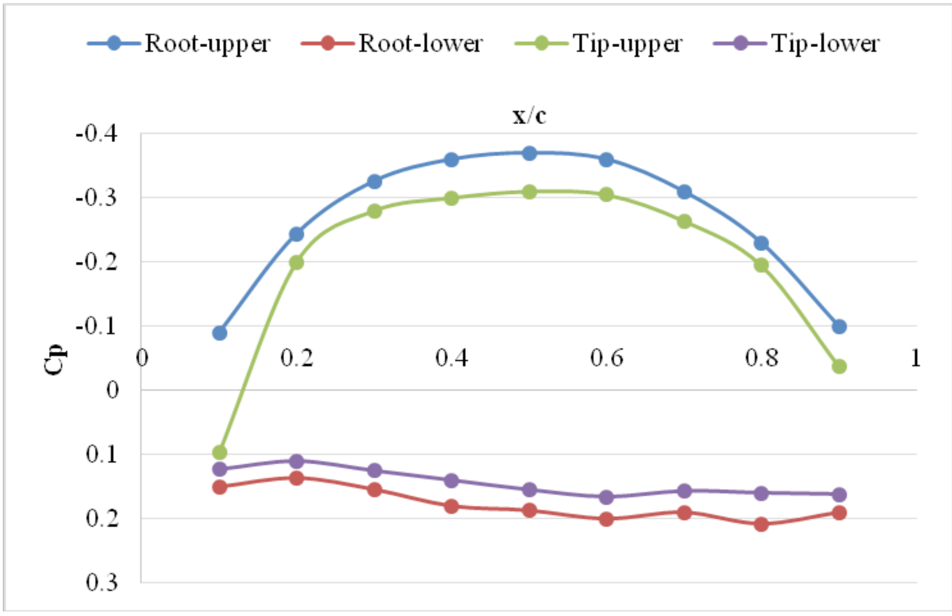

Figure 6.Pressure coefficients on upper and lower surfaces for the case of LIF=1.0 (Case of nil lift increment, starting point in design)

Conclusions

Minimum induced drag producing camber for its specified value along with lift constraint is established. The resulting profile is seen to produce increased negative pitching moment. Control over negative pitching moment is resulting in a non- reachable design solution. It is therefore advisable to use canard control for UAVs designed for high altitude long endurance compounded with high speeds.

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